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A METHOD FOR PREDICTING IGNITION ENERGY REQUIREMENTS OF PRACTICAL PROPELLANT SYSTEMS. PART III - ROCKETS

Prepared by:

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ABSTRACT: Basic ignition information was applied in correlating ignition data from 53 rockets of World War II and later vintage, ranging in area exposed to ignition products from 145 cm^2 to 320,000 cm^2. The dominant factor in setting ignition energy requirements for rockets was found to be, as for guns, the product of the area exposed to the ignition system products and the ignition energy per unit area of propellant.

The equations developed have relatively few variables, but a precision adequate for predicting ignition energy requirements in practical rocket ignition systems. Use of these equations in the design of efficient ignition systems is discussed.

CHEMISTRY RESEARCH DEPARTMENT
U. S. NAVAL ORDNANCE LABORATORY
White Oak, Silver Spring, Maryland

NAVORD Report 6134 2 February 1959

A scientific basis is desirable for designing ignition systems in order to reduce cost and effort associated with trial and error methods. This report provides a logical technique for designing rocket igniters. Attention of the engineer is called to Appendix VIII where a sample calculation is made. It is recommended that this method be tested in designing experimental ignition systems. This work was supported by Task NO 506/925/56015/01040.

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By direction

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NOMENCLATURE

A	Total area exposed to products of ignition system, cm^2
A_{ics}	Area of inhibited curved surface of grain, cm^2
A_{ie}	Area of inhibited ends of grain, cm^2
A_{ncs}	Area of convergent section of nozzle, cm^2
A_p	Port area; the free cross-sectional area in the rocket motor available for gas flow, cm^2
A_s	Area of burning surface of grain, cm^2
A_t	Nozzle throat area, cm^2
D	Diameter of propellant charge (in guns), cm
D_c	Inner diameter of rocket chamber, cm
D_g	Outer diameter of grain, cm
D_t	Nozzle throat diameter, cm
D_{ncs}	Largest diameter of nozzle convergent section, cm
J	Throat-to-port area ratio, dimensionless
K_I	Internal area ratio, equal to burning surface/port area, dimensionless
K_N	Nozzle area ratio, equal to burning surface/throat area, dimensionless
L	Length from midplane of vented section of primer to end of charge (in guns), cm
L_c	Length of straight section of chamber, cm
L_g	Length of grain, cm
N_{ie}	Number of inhibited ends of grain, dimensionless
P	Perimeter of burning surface of grain, cm
Q	Total energy supplied for ignition, cal
q_c	Experimental ignition energy per unit area of propellant (based on three millisecond ignition interval), cal/cm^2
q_r	Ignition energy per unit area supplied by rocket ignition system, cal/cm^2
r	Correlation coefficient, dimensionless
S_y	Standard error of estimate, dimensionless
V_f	Free volume in loaded chamber or cartridge case of gun

A METHOD FOR PREDICTING IGNITION ENERGY REQUIREMENTS OF PRACTICAL PROPELLANT SYSTEMS. PART III - ROCKETS

I. INTRODUCTION

In an attempt to develop a relation for predicting ignition energy requirements in practical systems, a correlating equation was developed for guns (1,2), using data from service-accepted rounds. The dominant correlating factor was the product of the total area exposed to the products of the ignition system and the ignition energy requirements per unit area of propellant. Three secondary factors were discovered: two involving the geometry of the round, and one, the degree of confinement of the ignition composition.

The correlation studies have been further extended to include rocket ignition systems. This paper presents the results obtained by analysis of data from service-accepted and advanced developmental rockets. A brief summary of this work has already been reported (2).

II. SCOPE

The principal rocket data (see Table I) are from 53 service-accepted or advanced developmental units varying in area exposed to ignition products, A, from 145 cm^2 to 320,000 cm^2. These include only those of World War II and later vintage. Literature references used in obtaining the necessary data are given in Tables II and III.

The rocket propellants were either double-base, or composite containing ammonium perchlorate or potassium perchlorate oxidizer.

Ignition and sustainer compositions used in these rockets (see Table II) included black powder, double-base propellant, seven different pyrotechnic compositions, and an ammonium-perchlorate-polysulfide rubber composite propellant.

Grain designs included the following:

1. Internal burning - These comprised about one-half of all grain designs considered, and were either case-bonded, or inhibited on the outside surface and obturated. The burning surface of these grains was in the form of a single cylindrical perforation, a single cylindrical perforation with slots, or a star, helical, conical, cruciform, or multi-perforated design.

2. External-internal burning (hollow cylinder) - About one-third of all grain designs had this geometry.

3. Internal burning shell with external-internal burning tube or external burning rod.

4. Cruciform.

5. End-burning (cigarette-burning).

III. TREATMENT OF DATA

A. Ignition Energy Supplied by Ignition Composition

The total energy, Q, liberated by the ignition composition is equal to the heat of explosion times the weight of the mixture.

Table II lists the procedures and literature references used in finding the heats of explosion, numerical values of the heats of explosion, and the composition of the various ignition mixtures.

In Table III the ignition mixtures used in the various rockets are given. The same references listed as pertaining to propellant and metal part dimensions supplied information on the weights of the ignition compositions.

B. Ignition Energy of Propellant (q_c)

Ignition energies experimentally determined in a locked-stroke compressor for a three millisecond ignition interval are given in Ref. 22 for representative propellant formulations. These values were used to estimate the ignition energy of a given propellant either by taking the ignition energy to be equal to that for the nearest chemically similar formulation, or by interpolating between the values for two chemically similar compositions.

C. Calculation of Area (A) Exposed to Products of Ignition System

The total area exposed to heat transfer from the products of the ignition system is composed of the following areas:

1. Burning surface of propellant grain.

2. Inhibited surface of propellant grain.

3. Metal Chamber.

4. Nozzle: Converging section of nozzle.

5. Miscellaneous metal parts other than 3 and 4, such as propellant supports, and trap plate.

6. Igniter assembly (neglected).

Calculation of these areas is detailed under the separate subheadings. Literature references used to supply the necessary information for the calculations are presented in Table III.

The relative importance of the different areas exposed to ignition products varied according to the charge design. For (1) internal-burning grains, not case-bonded; (2) external-internal burning grains; and (3) rod-and-shell type, the sum of the burning surface area plus the area of the straight section of the case plus the inhibited curved surface area of the propellant (zero for (2)) was 86 to 98% of the total area, A, with most values over 90%.

With case-bonded grains, where the products of ignition cannot travel between the case and the grain, the significant area was the burning surface, which made up 85 to 98% of the total area.

For cigarette-burning grains, the burning surface was a small percentage of the total area and the areas of the curved inhibited surface of the grain and the straight portion of the case were important. For the two cases treated in this report these surfaces constituted about 80% of the total area.

With the cruciform grain design the largest areas were the burning surface, straight portion of the case, and the inhibited area. In two British rockets (16) these areas were 97% of the total. In the "Tiny Tim," (8) having a charge consisting of four cruciform grains, the area of the supporting partitions was important, and the sum of the above areas was 64% of the total.

Burning Surface of Propellant Grain (A_s)

The burning surface was calculated by one of the following procedures:

1. Given: nozzle throat diameter and K_N, where $K_N = A_s/A_t$ = burning surface area/nozzle throat area. Nozzle throat area, A_t, was calculated. A_s was computed from

$$A_s = K_N A_t$$

2. Burning perimeter, P, was calculated from a scale or dimensioned drawing. Burning surface was calculated from

$$A_s = PL_g$$

where L_g = grain length

Uninhibited end areas must be added.

3. Burning surface area was given.

Method 1 was used in most of the calculations.

Area of Inhibited Surface of Propellant

Inhibited surface area, for purposes of calculation, could be divided into three parts: (1) inhibited end surfaces of the grain; (2) inhibited area on the curved surface of the grain; and (3) other inhibited areas such as strips of inhibitor on cruciform grains

1. Inhibited End Area (A_{ie}). Except for cigarette-burning grains,

$$A_{ie} = ((\pi/4) D_c^{\,2} - A_p) N_{ie}$$

where

D_c = inner diameter of chamber

A_p = port area; the free cross-sectional area in the rocket motor available for gas flow (see Section III-D for calculation of A_p)

N_{ie} = number of inhibited ends = two per grain, except for cigarette-burning grains, where N_{ie} = 1.

In the case of internal-burning, non-case-bonded grains, the above expression gives slightly high values of A, since the port area does not include the area of the annulus between the case and the inhibited outer surface of the propellant grain.

For a cigarette-burning grain,

$$A_{ie} = (\pi/4) D_g^{\,2}$$

where

D_g = diameter of grain.

2. Inhibited Area on Curved Surface of Grain (A_{ics}). The following methods were used:

a. If the grain was internal-burning and inhibited on the outside,

$$A_{ics} = \pi D_g L_g$$

b. If the grain was case-bonded,

$$A_{ics} = 0$$

3. Other Inhibited Areas. These were evaluated from dimensions which were given, or in one case, the area of inhibitor strips on a cruciform grain was estimated by dividing the inhibitor weight by the product of the thickness and density of the inhibitor.

Chamber

1. Head End. The area of this section was approximated by $(\pi/4)D_c^2$. This did not cause appreciable error, since the highest value of this term was 6.4% of the total area.

2. Straight Section. This area was equal to

$$\pi D_c \; L_c$$

where

L_c = the length of the straight section of the chamber.

Nozzle

The nozzle area exposed to the products of the ignition system was approximated and included only the convergent section. The half angle of the inlet cone was assumed to be 45 degrees. Since the nozzle areas evaluated were only 0.4 to 7% of A, with an average value equal to 2.8% of A, errors from this source should be small.

For rockets with one nozzle the largest diameter of the converging section, considering it to be a conical frustrum, was taken equal to the chamber diameter. This gave

$$A_{ncs} = \frac{(\pi/4)(D_c^2 - D_t^2)}{\sin 45^\circ} = 1.111 \; (D_c^2 - D_t^2)$$

where

A_{ncs} = area of converging section of nozzle.

D_t = diameter of nozzle throat.

In a multi-nozzled rocket, the largest diameter of the converging section (D_{cs}) was used, rather than the chamber diameter, so that for one nozzle

$$A_{ncs} = 1.111\ (D_{cs}^2 - D_t^2)$$
(Multi-nozzled rocket)

Miscellaneous Metal Parts

These have been neglected in most cases, except where the area of the grain supports was important. It is believed that no errors greater than 10% have been introduced as a result.

The area of the propellant trap is approximately one-half the cross-sectional area of the chamber or $(\pi/8)D_c^2$. By neglecting this area an error of about 0.5 to 2% in A is introduced.

D. Calculation of Port Area

Port area, A_p, was evaluated as follows:

1. If A_t and J were given, A_p was calculated from

$$A_p = A_t/J$$

where

J = throat-to-port area ratio.

2. J was calculated from

$$J = K_I/K_N$$

where

K_I = internal area ratio = burning surface/port area

K_N = burning surface/throat area

3. The end-grain area was calculated from a scale or dimensioned drawing. Other areas such as metal supports were calculated. The above areas were subtracted from $(\pi/4)D_c^2$ (the cross-sectional area of the case).

4. Port area was computed directly from a scale or dimensioned drawing.

5. For a cigarette-burning grain, the "port area" for the purposes of computing the secondary correlating factor was set equal to $(\pi/4)(D_c^2-D_g^2)$. This gave better results than using the

the actual port area, $(\pi/4)D_c^2$, when the inert cylindrical areas were included. This type of grain merits further study.

IV. CORRELATION TECHNIQUES

Preliminary plotting of Q vs. Aq_c indicated that Aq_c was a strong correlating factor (this proved to be the dominant factor as shown by Eq. 3). In order to determine secondary correlating factors, various parameters were chosen which seemed reasonable based on physical considerations. If a plot of q_r/q_c vs. a particular parameter showed some correlation, as judged by eye, the parameter was known to be a correlating factor.

Standard least-square techniques were used to derive the correlating equations. Scatter of the data points was evaluated by computing the standard error of estimate and correlation coefficient.

V. RESULTS

The logical first step was to test the dominance of Aq_c in determining the ignition energy requirements for rockets, since this product had been found dominant for guns (1,2). Figure 1 shows a plot of Q versus Aq_c (log-log). A least squares analysis of this data yielded

$$Q = 26.5\ (Aq_c)^{0.97} \tag{1}$$

$$r = 0.954$$

$$S_y = \pm\ 0.259 = \pm\ \log 1.82$$

Clearly, Aq_c is an important correlating factor.

$L_g/\sqrt{A_p}$ and A/A_pL_g, which are analogous to the secondary factors found for guns, were tested as secondary factors. No correlation was found for the first of these, but a fairly good correlation was obtained with A/A_pL_g. Other factors were tried. The best correlation found thus far was the ratio of two perimeters - one corresponding to the total length of exposed perimeter at a cross-section, and the other to the perimeter which a circle would have if equal in area to the port area. Using the port area perimeter in the numerator this ratio is

$$\sqrt{4\pi A_p}/(A/L_g)$$

Figure 2 shows a log-log plot of q_r/q_c versus the perimeter ratio where $q_r = Q/A$. A least squares analysis yielded

$$q_r/q_c = 56(L_g\sqrt{4\pi A_p}\ /A)^{0.59} \qquad (2)$$

$$r = 0.589$$

$$S_y = \pm\ 0.210 = \pm\ \log 1.62$$

Figure 3 shows a log-log plot of Q versus

$$Aq_c(L_g\sqrt{4\pi A_p}/A)^{0.59}$$

A least squares analysis of this plot yielded

$$Q = 38\left[(Aq_c)(L_g\sqrt{4\pi A_p}/A)^{0.59}\right]^{1.06} \qquad (3)$$

$$r = 0.972$$

$$S_y = \pm\ 0.205 = \pm\ \log 1.60$$

From the above equations it can be clearly seen that Aq_c is the dominant factor and $\sqrt{4\pi A_p}/(A/L_g)$ is a secondary correlation factor.

Analysis of experimental firings of rockets for upper and lower limits to the ignition energy requirements has, as yet, been limited. Examination of data by Warren and Cronhardt (20) indicated that upper and lower limits could vary by a factor of two (in one case a factor of 4) with other variables constant. Reference 23 gave an example in which the ignition system for the "Tiny Tim" rocket was altered so that the ignition charge could be reduced by a factor of five and still give satisfactory operation. Calculations were made on this rocket (Reference 23 gave figures in grains that should have been grams (24)). The large charge was at + 1.57 S_y from the predicted charge and the small charge was at -1.89 S_y.

VI. DISCUSSION OF RESULTS

A. Correlation Studies

By combining basic ignition energy information with data on 53 rockets (service-accepted, advanced developmental, and a few experimental), an equation involving only a few variables has been obtained for predicting ignition energy requirements of practical rockets. Since the rockets included had internally exposed areas ranging from 145 to 320,000 cm^2 and included a variety of ignition system designs and ignition compounds (pyrotechnic, black powder and double-base), an assortment of propellant compositions and shapes, and a number of design sources, the equation should be applicable to a wide range of rockets. Although this work is in an earlier stage than that for guns, the precision obtained is comparable.

The dominant factor in determining the ignition energy requirements, as in the case of gun ignition systems (1,2), is Aq_c, the product of total area exposed to the products of the ignition system and the ignition energy requirements per unit area of propellant. There is a significant difference between rockets and guns in the correlation of Q with Aq_c. Generally q_r/q_c is higher than q_g/q_c. Part of this difference is probably due to using a q_c for a three millisecond energy transfer time for both guns and rockets, though ignition intervals for rockets average considerably longer than for guns. In addition, smaller ratios of area to free volume, lower ignition pressures, and loss of ignition products through the nozzle (in some cases) prevent efficient use of the energy in rocket ignition systems. There is some indication that the difference in ignition intervals may be the largest of these factors when nozzles are obturated during ignition. All of these variables are interconnected, however, and further analysis may help to clarify their relative influence.

The secondary factor, $L_g \sqrt{4\pi A_p}/A$ may be considered as a measure of radial complexity (inverse relationship). It is related to V_f/A for guns, and both have a similar influence. Differences in radial complexity thus seem most important in rockets; whereas, differences in radial and differences in longitudinal complexity seem important in guns. Perhaps further analysis will locate a more fundamental factor than the perimeter ratio used.

For a cigarette-burning grain $L_g \sqrt{4\pi A_p}/A$ is not a measure of radial complexity. An equivalent term should be developed to handle this case. Since only two cigarette-burning grains have been considered out of a total of 53 rockets, the correlations should not have been appreciably affected.

It is of interest that the accuracy of prediction of Q for rockets (judging by the correlation coefficient and standard error of estimate) is approximately the same as for guns. For guns, $S_y = \pm \log 1.58$ and $r = 0.977$. Likewise, the uncertainty seems to be largely due to the range of ignition energies which will give satisfactory ignition plus the range in effectiveness of the various igniter designs in delivering the energy. Calculations on the "Tiny Tim," for which the designers made considerable effort by bot[illegible] design and charge changes to lower the charge weight, showed the [illegible] original charge was at + 1.57 S_y and the final charg[illegible] -1.89 S_y from the value predicted by Eq. 3. When this is compared with the predicted upper and lower limits of $\pm$ 2 S_y, the results are striking.

The ignition energy equation (Eq. 3) is probably applicable to a greater diversity of ignition compositions than the corresponding equation for guns, because a greater variety of igniter compositions (black powder, pyrotechnic mixtures, and double-base powders) were included in the data from which the rocket equation was derived.

The correlations can probably be further improved by considering the effect of heat-transfer rate. Calculations from experimental data on M-2 propellant showed that ignition energy requirements decreased as heat flux increased (22). The rate of cooling of the ignition products in the annular space between the grain and the case is no doubt higher than in the grain perforation, and allowance should be made for this. It would seem an external-burning grain should be treated differently from an externally inhibited and obturated one. In the first case the external surface not only acts as a heat sink, but must also be ignited. In the second case, the surface acts only as a heat sink, and, depending on the location of the obturation, ignition may hava already occurred before this annular volume is filled.

The characteristics of the ignition system such as degree of confinement of the ignition mixture (found to be important for gun ignition systems) and the composition of the ignition mixture as it affects flame temperature and the solids/gas ratio in the ignition products influence the rate of heat transfer and should therefore be examined. Ignition delay time might also be useful as a correlating parameter.

B. Application of Equations

Assume that all ignition systems will give satisfactory ignition over a range of ignition energies of two for a given weapon. The center line value predicted by Eq. 3 will then give satisfactory results with all ignition systems which have lower limits above -1.47 S_y (at Q/2) and upper limits below + 1.47 S_y (at 2Q). If we make the additional assumption that the upper and lower limits of ignition systems are distributed similarly to the scatter in data, the centerline value would give satisfactory results with 86% of the ignition systems used. We might replace the second assumption by the more pessimistic one that all data points were at the center of the possible range for the given ignition system. The probability of getting satisfactory ignition with the predicted centerline value is then reduced to 54%.

With the first pair of assumptions above, a satisfactory charge for 99% of the ignition systems could be obtained by a two-trial method. First a value of one standard error of estimate below the centerline value would be used and, if unsuccessful (should be successful or under-ignition), a value one standard error of estimate above the centerline value would be tried. This method would also give some overlap in the region in which igniters have the highest probability of having an operating range. Using the more pessimistic assumptions a probability of 92% is still obtained.

Another method of approach (making no assumptions) would be to use the predicted value and modify or change the igniter until satisfactory operation was obtained. There would seem to be a high probability of success for this method by making modest changes in conventional ignition systems.

The problem might be to design as small an ignition system as reasonably possible. In this case, a value about 2 S_y below the predicted centerline value would probably provide a difficult enough problem and a value 3 S_y below would probably give an extremely difficult design problem.

VII. SUMMARY AND CONCLUSIONS

Basic ignition information has been applied to correlating ignition data from a large number of rockets ranging in area exposed to ignition products from 145 cm^2 to 320,000 cm^2.

The dominant correlating factor is Aq_c, the product of total area exposed to the products of the ignition system and the ignition energy requirements per unit area of propellant.

The secondary correlating factor is $L_g\sqrt{4\pi A_p}/A$ which is the ratio of the perimeter of a circle equal in area to the port area to the total perimeter exposed at a cross-section.

The correlation (Eq. 3) is probably applicable to a greater diversity of igniter compositions than the correlation for gun ignition systems (1,2).

The correlation could probably be further improved by considering the effect of heat-transfer rate from the ignition products to the propellant, considering such variables as flame temperature of the ignition mixture and gas/solids ratio in the ignition products.

The effect of the degree of confinement of the ignition mixture (found to be important in guns) should be examined.

The relative weighting of the various areas exposed to the ignition products needs further study, particularly in the case of non-case bonded, internal-burning grains. The weighting of the area of the annulus between the grain and the case wall, compared to the area of the internal perforation, should especially be considered.

VIII. APPENDIX

Sample Calculation of Weight of Igniter Composition

The problem is to calculate the weight of igniter composition required for the satisfactory ignition of a solid-propellant rocket. The total energy required for ignition is given by Eq. 3 as

$$Q = 38\left(Aq_c\left[L_g\sqrt{4\pi A_p}/A\right]^{0.59}\right)^{1.06} \quad \text{(A-1)}$$

where

A = total area exposed to products of igniter system, cm^2

A_p = port area, cm^2

L_g = length of grain, cm

q_c = Experimental ignition energy per unit area of propellant (based on three millisecond ignition interval), cal/cm^2

Q = Total energy supplied for ignition, cal

The weight of igniter composition is found from

$$W = Q/\Delta H \quad \text{(A-2)}$$

where

ΔH = heat of explosion of igniter composition, cal

W = weight of igniter composition, gm

DATA

Consider an arbitrary rocket motor having the characteristics shown below:*

Dimensions of Metal Parts

Chamber:	
Length, L_c	52 in.
Inner diameter, D_c	5.50 in.
Nozzle:	
Throat diameter, D_t	1.375 in.

*The presentation of data here is similar to the form employed in Ref. 8.

Propellant Data

Propellant designation	OGK
Charge design	Star-perforated (internal burning)
Inhibitor location	On outer diameter and ends

Propellant dimensions (including inhibitor):

Number of grains	One
Length, L_g	51 in.
Diameter, D_g	5.45 in.

Internal ballistics:

Initial burning surface/throat area, K_N	430
Port area/throat area, 1/J	2.5

CALCULATION OF AREA (A) EXPOSED TO PRODUCTS OF IGNITION SYSTEM

$$A = A_{ncs} + A_s + A_{ics} + A_{ie} + \text{chamber areas} \qquad \text{(A-3)}$$

where

A_{ncs} = area of convergent section of nozzle

A_s = area of burning surface of propellant grain

A_{ics} = area of inhibited curved surface of grain

A_{ie} = area of inhibited ends of grain

Chamber areas = area of straight section of chamber plus area of head end.

Nozzle throat area, $A_t = \frac{\pi}{4}(D_t)^2$ (A-4)

$$A_t = (\pi/4)(1.375)^2 = 1.485 \text{ in}^2$$

$$A_{ncs} = (1.111)(D_c{}^2 - D_t{}^2) \quad \text{(see section III-C ... derivation)} \qquad \text{(A-5)}$$

$$A_{ncs} = (1.111)\left[(5.50)^2 - (1.375)^2\right] = 31.51 \text{ in}^2$$

$$A_s = K_N A_t \qquad \text{(A-6)}$$

$$A_s = (430)(1.485) = 638.6 \text{ in}^2$$

$$A_{ics} = \pi D_g L_g \qquad \text{(A-7)}$$

$$A_{ics} = (\pi)(5.45)(51) = 873.2 \text{ in}^2$$

$$\text{Port area, } A_p = A_t/J \qquad \text{(A-8)}$$

$$A_p = (1.485)(2.5) = 3.713 \text{ in}^2$$

$$A_{ie} \cong \left[(\pi/4)(D_c{}^2) - A_p\right] N_{ie} \qquad \text{(A-9)}$$

where

N_{ie} = number of inhibited ends = 2

$$A_{ie} \cong \left[(\pi/4)(5.50)^2 - 3.713\right]\left[2\right]$$

$$A_{ie} \cong 40.09 \text{ in}^2$$

Equation A-9 was used for convenience in the compilation of some of the data in this report. It is correct for an internal-external burning grain, and is a good approximation in this case (internal-burning grain), since the clearance between the inhibited grain and the chamber wall is small.

Chamber Areas

1. The head-end area is approximated by

$$(\pi/4)(D_c)^2 = (\pi/4)(5.50)^2 = 23.76 \text{ in}^2 \qquad \text{(A-10)}$$

2. The area of the straight section is

$$\pi D_c L_c = (\pi)(5.50)(52) = 898.5 \text{ in}^2 \qquad \text{(A-11)}$$

Total Area, A

From Eq. A-3,

$$A = 31.51 + 638.6 + 873.2 + 40.09 + 23.76 + 898.5$$

$$A = 2506 \text{ in}^2$$

or

$$A = (2506 \text{ in}^2)(6.452 \text{ cm}^2/\text{in}^2)$$

$$A = 16{,}170 \text{ cm}^2$$

SECONDARY CORRELATION FACTOR, $L_g\sqrt{4\pi A_p}/A$

$$L_g\sqrt{4\pi A_p}/A = (51)\sqrt{(4\pi)(3.713)}/2506$$
$$= (51)\sqrt{46.66}/2474$$
$$= (51)(6.831)/2506$$
$$L_g\sqrt{4\pi A_p}/A = 0.1390 \text{ (dimensionless)}$$

TOTAL ENERGY REQUIREMENT, Q

q_c for OGK = 0.361 cal/cm^2 from Ref. 22

Using Eq. A-1 (Eq. 3),

$$Q = 38\left[(16,170)(0.361)(0.1390)^{0.59}\right]^{1.06}$$
$$= 38\left[(16,170)(0.361)(0.3122)\right]^{1.06}$$
$$= 38\ (1822)^{1.06}$$
$$= (38)(2859)$$
$$Q = 109,000 \text{ cal}$$

Q can also be found from Fig. 3. The abscissa is

$$\left[Aq_c\right]\left[L_g\sqrt{4\pi A_p}/A\right]^{0.59}$$
$$= (16,170)(0.361)(0.1390)^{0.59}$$
$$= 1822$$

From Fig. 3,

$$Q = 109,000 \text{ cal}$$

WEIGHT OF IGNITER COMPOSITION, W

Theoretical weight

From Eq. A-2,

$$W = 109,000/\Delta H$$

If black powder is used in the igniter,

$$\Delta H = 698.8 \text{ cal/gm}$$

and

$$W = 109{,}000/698.8$$

$$W = 156 \text{ gm}$$

This theoretical value of W, based on the value of Q given by Eq. A-1 (Eq. 3), is estimated to give satisfactory results with 55% to 87% of all ignition systems, depending on the assumptions made (see Section VII).

Two-Trial Procedure

As discussed in Section VII, a two-trial procedure can be used, where a value of Q one standard error of estimate below the theoretical value is used, and if unsuccessful (under-ignition), a value one standard error of estimate above the theoretical value is tried. A satisfactory charge for as many as 99% of all ignition systems should be found in this way.

The standard error of estimate, S_y, for Eq. A-1 (Eq. 3) is log 1.60. Thus the lower value of W to be tried would be

$$156/1.60 = 97.5 \text{ gm}$$

If this were unsuccessful, the upper value would be tried, equal to

$$(156)(1.60) = 250 \text{ gm}$$

TABLE I

SUMMARY OF DATA

	Rocket*	Q x 10^{-3} (cal)	A x 10^{-3} (cm^2)	q_r/q_c	q_c (cal/cm^2)	$\frac{L_g\sqrt{4\pi A_p}}{A}$	Q/Q**
1.	2.5" Mk 1	1.74	0.424	16.9	0.243	0.205	0.911
2.	2.5" Mk 6	1.74	0.564	12.7	0.243	0.218	0.647
3.	VFR CIT (200'/sec)	1.74	0.318	22.6	0.243	0.186	1.31
4.	1" Ducted Rocket	1.95	0.145	46.2	0.290	0.410	1.71
5.	J, 0.6-ES-200, T9E2	3.49	0.356	35.1	0.280	0.305	1.48
6.	15" Grain	3.91	1.91	7.34	0.280	0.162	0.416
7.	11.25" Grain	3.91	1.50	9.34	0.280	0.154	0.552
8.	7.5" Grain	3.91	1.09	12.8	0.280	0.141	0.820
9.	J, 1.134-KS-449, T2008	5.38	0.824	22.5	0.290	0.692	0.540
10.	2.75" AEROFLAX "K"	7.50	1.75	14.3	0.300	0.117	0.996
11.	4.5" BR	8.24	1.19	28.4	0.243	0.215	1.40
12.	7.2" Mk 6	8.24	1.20	28.3	0.243	0.199	1.46
13.	2.25" AEROSCAR Model 2	9.08	1.63	18.5	0.300	0.233	0.840
14.	7.2" VAR (200'/sec)	14.0	1.60	35.9	0.243	0.221	1.72
15.	7.2" VAR (300'/sec)	14.0	2.16	26.6	0.243	0.252	1.15
16.	6 Point Star	17.5	3.46	17.4	0.290	0.373	0.566
17.	Tail Rocket U 2 in.	18.8	2.12	37.1	0.240	0.311	1.40
18.	Target Flare	19.6	1.61	50.9	0.240	0.208	2.52
19.	J, 1-KS-2800, X106A1	21.0	4.35	16.1	0.300	0.198	0.757
20.	J, 0.7-ES-2650, M3E1	21.0	4.94	15.2	0.280	0.126	0.950
21.	J, 12-KS-250, T43	21.0	3.77	18.5	0.300	0.133	1.14
22.	Helical Grain	21.0	2.64	27.5	0.290	0.348	0.946
23.	7.2" CWR-N	24.4	3.24	31.0	0.243	0.218	1.43
24.	Cruciform 11	25.1	6.76	15.5	0.240	0.240	0.644
25.	Cruciform 12-1/2	25.1	6.74	15.5	0.240	0.208	0.707
26.	T.R. U 3 in.	26.7	5.93	18.8	0.240	0.244	0.778
27.	Conical Grain	28.0	1.91	50.3	0.290	0.901	0.979
28.	J, 8.5-DS-1000, X221A1	29.2	7.71	10.5	0.361	0.0511	1.11
29.	J, 1.8-DS-7000, X205B3	34.9	23.7	4.08	0.361	0.102	0.262
30.	J, Mk 2, Mod 3	34.9	10.7	10.9	0.300	0.137	0.616

TABLE I (Cont.)

SUMMARY OF DATA

	Rocket*	Q x 10^{-3} (cal)	A x 10^{-3} (cm^2)	q_r/q_c	q_c (cal/cm^2)	$\frac{L_g\sqrt{4\pi A_p}}{A}$	Q/Q**
31.	J, 6-KS-3000, T40	35.9	6.42	19.3	0.290	0.299	0.693
32.	J, 0.88-KS-3350	40.0	2.89	47.7	0.290	0.739	1.02
33.	J, 5.4-KS-1700	40.0	3.77	36.6	0.290	0.457	1.04
34.	J. 1.3-KS-4600, M58	41.3	4.17	34.1	0.290	0.364	1.11
35.	5.0" AEROHVAR Model 1	41.9	8.92	15.7	0.300	0.248	0.619
36.	Tail Rocket U 5 in.	47.1	7.57	25.9	0.240	0.150	1.44
37.	J, Mk 4, Model 2	52.1	12.1	12.0	0.361	0.0546	1.18
38.	J, 14-DS-1000,X209B2	66.1	12.5	14.7	0.361	0.737	1.20
39.	J, 2.2-KS-11000,X102F2	80.4	16.0	16.8	0.300	0.180	0.780
40.	2.5-ES-8400,B.S. 1.2	90.3	10.5	35.9	0.240	0.550	0.868
41.	J, 2.5-KS-18000,X103C10	126	26.7	15.7	0.300	0.159	0.768
42.	J, 1.8-KS-7800,X113C3	184	17.1	35.9	0.300	0.151	1.86
43.	J,30-DS-4000,T34	210	58.2	9.98	0.361	0.0750	0.739
44.	J, Mk 7, Model 1	228	24.6	32.0	0.290	0.139	1.71
45.	J, 3-DS-49000,X201C1	867	140	17.2	0.361	0.127	0.867
46.	J, 2.4-ES-57000,XM13	951	175	18.1	0.300	0.115	0.974
47.	J,2.2-KS-33000,X105G2	1000	50.7	65.7	0.300	0.250	2.34
48.	J,4-DS-105000,X202C10	1050	307	9.47	0.361	0.134	0.443
49.	J,4.1-DS-111000,X230A3	1290	263	13.6	0.361	0.0827	0.866
50.	J,2.7-KS-98000,XM16	1520	83.9	62.6	0.290	0.295	1.93
51.	J,4-DS-80000,X202A1	2340	236	27.4	0.361	0.131	1.32
52.	J,40-NS-4500,X118A1	2760	95.8	95.9	0.300	0.226	3.50
53.	J,4-DS-105000,X202C3	3100	320	26.8	0.361	0.133	1.26

* J = JATO
** Calculated from Eq. 3

TABLE II

HEATS OF EXPLOSION FOR IGNITION MIXTURES

Designation	Composition	Heat of Explosion (cal/gm)	Methods and Sources* Used in Finding Heat of Explosion
1. Black Powder	73.4% KNO_3, 16.4% C, 10.2% S	698.8	(3) △a
2. T-47 Igniter Composition (X-225) △1	72.3% $KClO_4$, 14.8% Ti, 6.9% B	1691	(4) △a, (5) △d
3. T-46 Igniter Composition △1	73.6% $KClO_4$, 20.4% B, 6% polyisobutylene	1650	(6) △d, (7) △b
4. △2	48.1% $KClO_4$, 48.1% Mg, 3.8% polyisobutylene	2000	(8) △d, (9) △b
5. ALCLO △3	64% $KClO_4$, 35% Al, 1% binder	2500	(10) △a, △d
6. △1	57.8% KNO_3, 34.8% Mg, 7.4% Accroides gum	1621	(11) △a, △d
7. Trench Mortar Sheet △4	Same as M8 or M9 double-base propellant	1270	(12) △d, (13) △a, △d
8. E-4 Igniter Comp.: TRX 126B △1	70.7% NH_4ClO_4, 21.5% LP-33 △5, 7.8% other ingredients	1160	(14) △d, (13) △b
8. E-4 Igniter Comp.: "2A" pellets △6	70.7% KNO_3, 23.6% B, 5.7% "Laminac" binder △7	1700	(15) △d, (14) △b, (7) △b

TABLE II (Cont.)

HEATS OF EXPLOSION FOR IGNITION MIXTURES

Designation	Composition	Heat of Explosion (cal/gm)	Methods and Sources* Used in Finding Heat of Explosion
9. JPN /8	Double-base composition	1175	(13) /a, /d
10. SR 371, 371A, or 371C /9	50% KNO_3, 42% Mg, 8% accroides gum	1570	(16) /d, (11) /b, (17)/c

* Indicated by numbers in parentheses.

/a Reference gave heat of explosion.

/b Reference supplied data on heat of explosion of chemically similar composition.

/c Reference supplied thermodynamic data used to calculate heat of explosion.

/d Reference supplied data on composition of ignition mixture.

/1 Made by Thiokol Chemical Corporation.
/2 Developed by Hughes Aircraft Company.
/3 Made by Aerojet-General Corporation.
/4 Made by Picatinny Arsenal.
/5 Liquid polysulfide synthetic rubber polymer manufactured by Thiokol Chemical Corp.
/6 Made by U. S. Flare Corporation.
/7 Liquid thermosetting polyester resin made by American Cyanamid Company.
/8 Made by U. S. Naval Ordnance Test Station.
/9 British igniter Composition.

TABLE III

SOURCES OF DATA

Rocket /1	Ignition Mixture Number /2	Literature References Used to Obtain Propellant and Metal Part Dimensions
1	1	(18)
2	1	(18)
3	1	(18)
4	2	(19)
5	1	(8)
6	1	(20)
7	1	(20)
8	1	(20)
9	8	(8)
10	5	(8)
11	1	(18)
12	1	(18)
13	1	(8)
14	1	(18)
15	1	(18)
16	1	(21)
17	10	(16)
18	10	(16)
19	1	(8)
20	1	(8)
21	1	(8)
22	1	(21)
23	1	(18)
24	10	(16)
25	10	(16)
26	10	(16)
27	1	(21)
28	1,9	(8)
29	1	(8)
30	1	(8)
31	1,6	(8)
32	4	(8)
33	4	(8)
34	3	(8)
35	1	(8)

TABLE III (Cont.)

SOURCES OF DATA

Rocket /1	Ignition Mixture Number /2	Literature References Used to Obtain Propellant and Metal Part Dimensions
36	10	(16)
37	1,7	(8)
38	1,7	(8)
39	1	(8)
40	10	(8)
41	1	(8)
42	5	(8)
43	1	(8)
44	1,5	(8)
45	1,9	(8)
46	1	(8)
47	5	(8)
48	1	(8)
49	1,9	(8)
50	2	(8)
51	1,7	(8)
52	1,5	(8)
53	1,7	(8)

/1 The tabulated numbers refer to the rockets listed in Table I.

/2 See Table II.

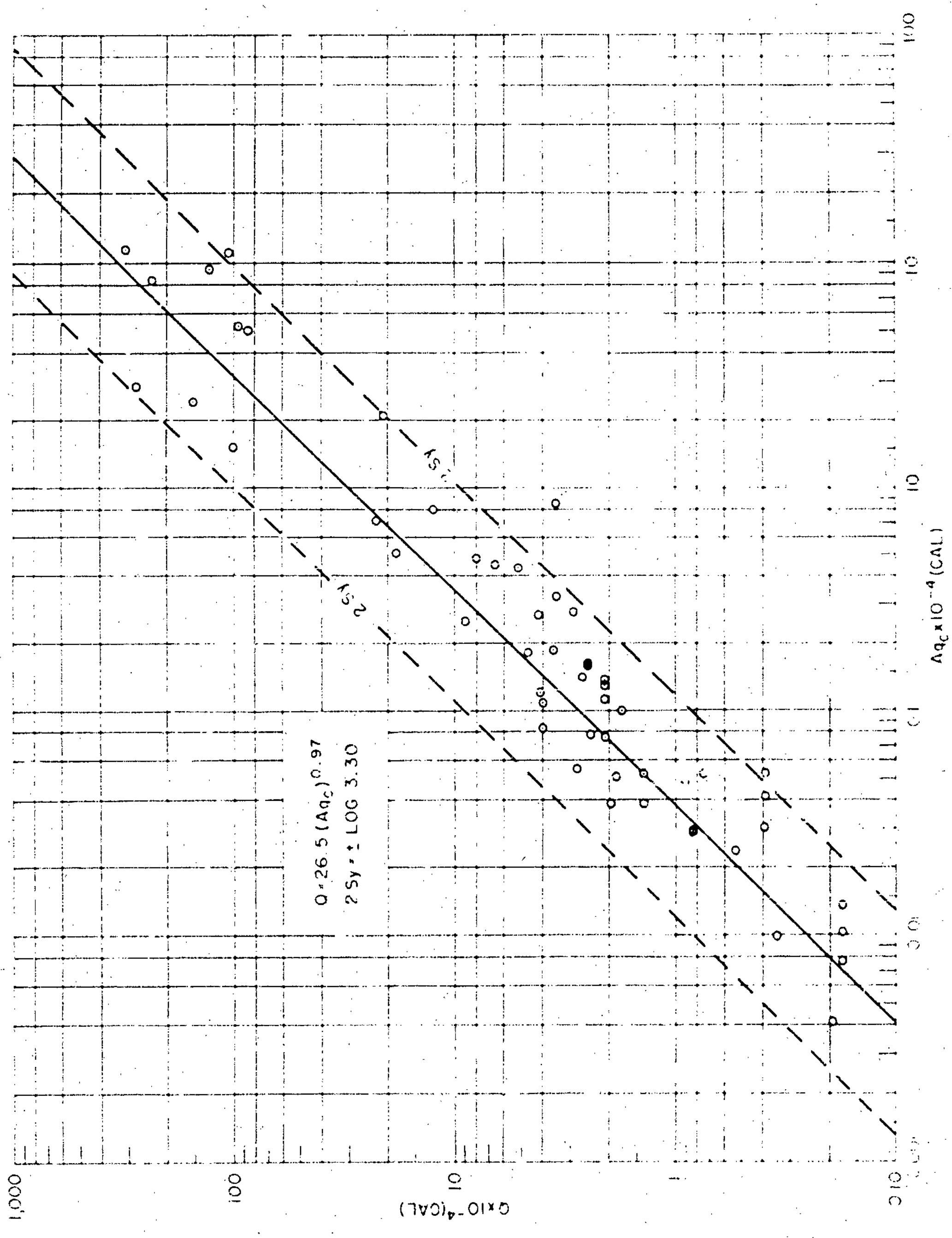

FIG 1 TOTAL IGNITION ENERGY VS DOMINANT CORRELATING FACTOR FOR ROCKETS

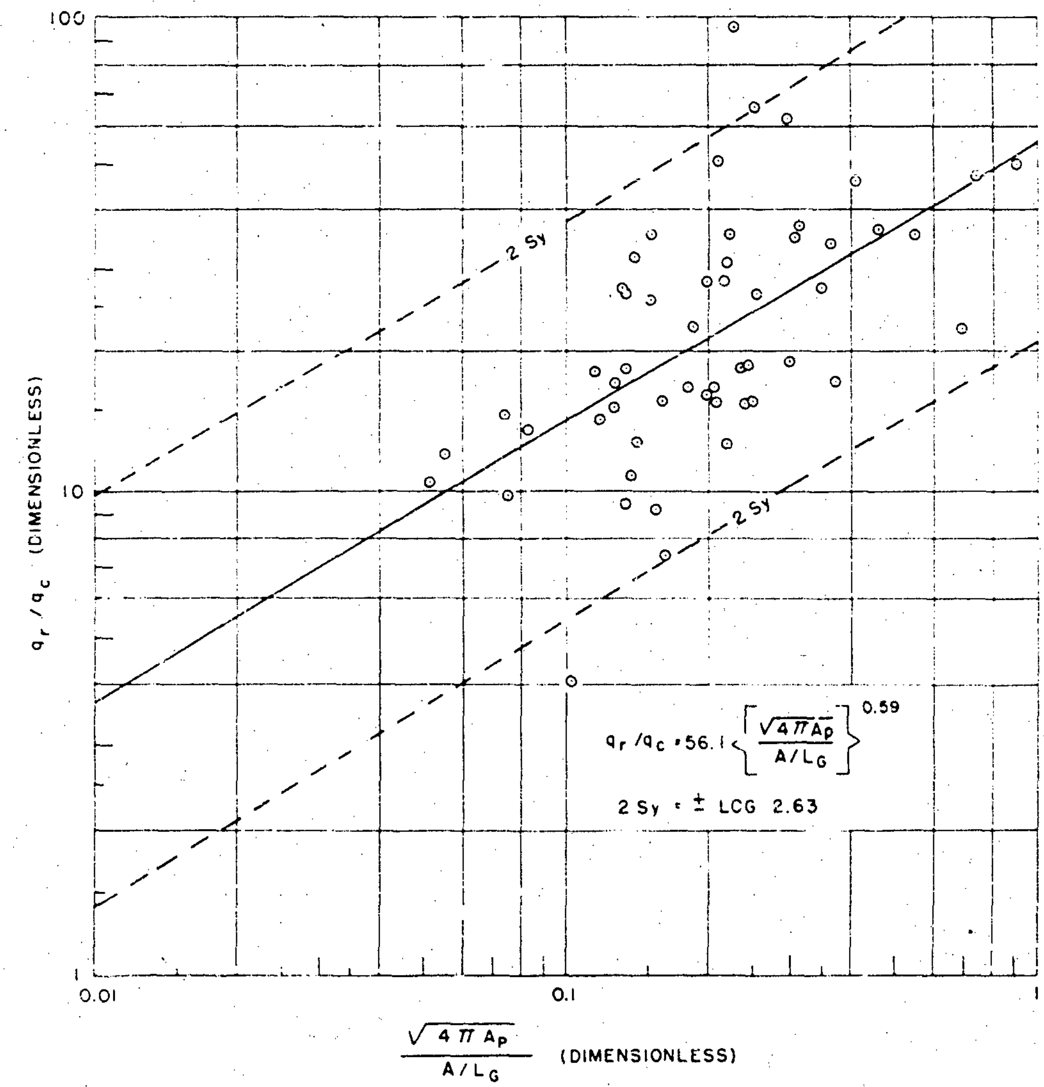

FIG. 2 q_r/q_c VS SECONDARY CORRELATING FACTOR FOR ROCKETS

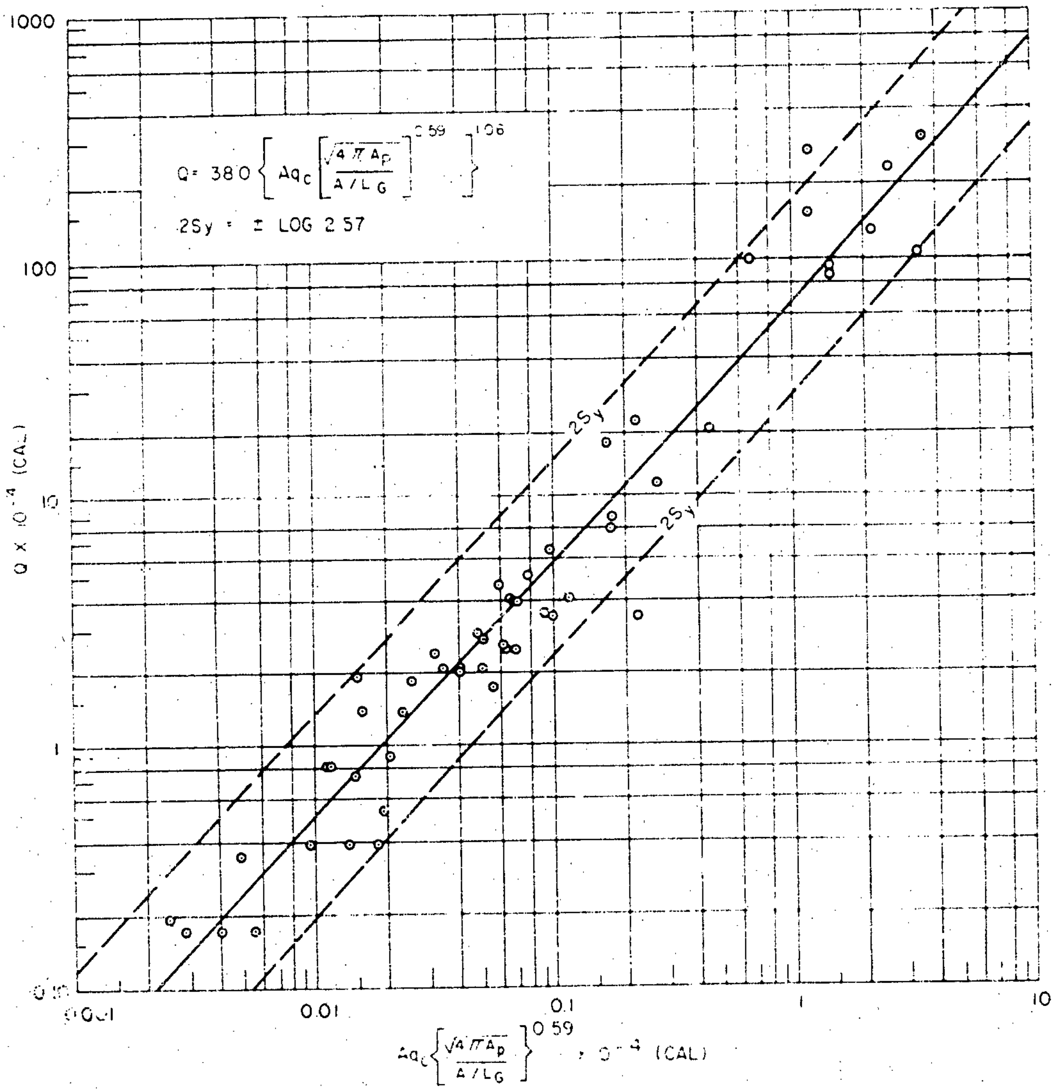

FIG. 3 CORRELATION OF TOTAL IGNITION ENERGY FOR ROCKETS

BIBLIOGRAPHY

1. Bryan, G. J., and Lawrence, E. K., "A Method for Predicting Ignition Energy Requirements of Practical Propellant Systems. Part II - Guns", NOL NAVORD 5750, 18 December 195. . CONFIDENTIAL.

2. Bryan, G. J., and Lawrence, E. K., "Application of Fundamental Data to Gun and Rocket Ignition", Bulletin of the Second Symposium on Solid Propellant Ignition, Vol. I, SPIA (1956). CONFIDENTIAL.

3. Bryan, G. J., and Edwards, G. D., "A Method for Predicting Ignition Energy Requirements of Practical Propellant Systems. Part I", NOL NAVORD 4189, 20 December 1955. CONFIDENTIAL.

4. Private Communication from F. P. Anderson, Thiokol Chemical Corporation, Redstone Division, to E. . Lawrence, Naval Ordnance Laboratory, 18 May 1956. CONFIDENTIAL.

5. Thiokol Chemical Corporation, Redstone Division, "Development of the T-47 Rolled Tube Igniter for the LOKI Rocket Motor", Report No. 45-54, Redstone Arsenal, Huntsville, Alabama, September 1954. CONFIDENTIAL.

6. Thiokol Chemical Corporation, Redstone Division, "Model Specification Rocket Motor, Solid Propellant T58 El, 1.3-KS-4600 MS-RE-2, Report No. 33-54, Redstone Arsenal, Huntsville, Alabama, June 1954. CONFIDENTIAL.

7. Thiokol Chemical Corporation, Redstone Division, "Progress Report. Ignition of Polysulfide-Perchlorate and Related Rocket Propellants", Report No. 40-53, Redstone Arsenal, Huntsville, Alabama, October 1953. CONFIDENTIAL.

8. Solid Propellant Information Agency, "Jato Manual SPIA/M1", Vol. II, The Johns Hopkins University Applied Physics Laboratory, Silver Spring, Maryland, April 1956. CONFIDENTIAL.

9. Thiokol Chemical Corporation, Redstone Division, Quarterly Progress Report, No. 41-54, (April through June 1954), Redstone Arsenal, Huntsville, Alabama. CONFIDENTIAL.

10. Bressler, M., "Development of Igniter Materials and Igniter Design Methods at the Aerojet-General Corporation", Bulletin of Twelfth Meeting of the JANAF Solid Propellant Group, Vol. II, SPIA (1956). CONFIDENTIAL.

11. Thiokol Chemical Corporation, Redstone Division, "Progress Report. Development of Metal Powder Oxidizer Compositions for Ignition of Polysulfide-Perchlorate Propellants", Report No. 9-52, Redstone Arsenal, Huntsville, Alabama, March 1952. CONFIDENTIAL

12. Department of the Army, "Military Explosives", Department of the Army Technical Manual TM 9-1910, Washington, U. S. Government Printing Office, April 1955. UNCLASSIFIED.

13. Solid Propellant Information Agency, "Propellant Powder Manual SPIA/M2", The Johns Hopkins University Applied Physics Laboratory, Silver Spring, Maryland, January 1957. CONFIDENTIAL.

14. Thiokol Chemical Corporation, Redstone Division, Quarterly Progress Report, No. 1-56, (October through December 1955), Redstone Arsenal, Huntsville, Alabama. CONFIDENTIAL.

15. Thiokol Chemical Corporation, Redstone Division, "Development of Improved Ignition System for the T2008 Rocket Motor", Report No. 36-55, Redstone Arsenal, Huntsville, Alabama, October 1955. CONFIDENTIAL.

16. James, F., "Methods of Igniting the Propellant in Rockets", P.D.E. Report No. 1942/58 (British), August 1942. UNCLASSIFIED.

17. Rossini, F. D., et al, "Selected Values of Chemical Thermodynamic Properties", Circular of the National Bureau of Standards 500, Washington, U. S. Government Printing Office, February 1952. UNCLASSIFIED.

18. OSRD, National Defense Research Committee, California Institute of Technology, "Ammunition Catalog CIT Rockets", Pasadena, California, 10 August 1943 (Revised as of 10 February 1944). CONFIDENTIAL.

19. Thiokol Chemical Corporation, Redstone Division, "Final Report Solid Propellant Ducted Rocket", Report No. 25-55, Redstone Arsenal, Huntsville, Alabama, July 1955. CONFIDENTIAL.

20. Cronhardt, R. S., and Warren, F. A., "Ignition of Rocket Motors", NOTS 453, NAVORD 1925 (1951). CONFIDENTIAL.

21. Thiokol Chemical Corporation, Redstone Division, "Final Report on Development of Jatos 5-KS-900, T44 and 2-KS-2000, T45", Report No. 4-50, Redstone Arsenal, Huntsville, Alabama, August 1950. CONFIDENTIAL.

22. Bryan, G. J., "Ignition of Propellants by Hot Gases. Part III. Ignition Energy Requirements and Heat Transfer under Transient Conditions", NOL NAVORD 3818, 20 October 1955. CONFIDENTIAL.

23. Burchard, John E., "Science in World War II. Rockets, Guns, and Targets", p. 163; Little, Brown, and Co., Boston (1948). UNCLASSIFIED.

24. Department of the Army, "Rockets", Department of the Army Technical Manual TM 9-1950, Washington, U. S. Government Printing Office, July 1950. RESTRICTED.

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